

FUTURE EUROPEAN REUSABLE BOOSTER STAGES: EVALUATION OF VTHL AND VTVL RETURN METHODS

Jascha Wilken*, Sven Stappert, Leonid Bussler, Martin Sippel, Etienne Dumont

Department of Space Launcher Systems Analysis (SART), Institute of Space Systems, German Aerospace Center (DLR), Robert Hooke Straße 7, 28359 Bremen, Germany

* Corresponding Author: jascha.wilken@dlr.de

Abstract

Reusability of launch systems will strongly impact the launch service market if certain characteristics, such as sufficient reliability and low refurbishment costs, can be achieved. The German Aerospace Center (DLR) is performing a systematic investigation of return methods for a reusable booster stage of a future European launch vehicle. This launcher shall be able to transport 7 t to a geostationary transfer orbit (GTO), launching from the European spaceport in Kourou. The final goal is the determination of the impact of the different return methods on a technical, operational and economical level and the assessment of their relevance for a future European launch system. Compared to previous work presented at the IAC 2017 [3], this paper includes winged vertical take-off and horizontal landing (VTHL) as well as non-winged vertical take-off and vertical landing (VTVL) configurations.

The preliminary results of a first design phase employing structural indexes derived from existing stages were used to narrow down the field of potential designs, especially with regard to the rocket staging. The selected launchers were modelled in more detail, including the preliminary design of major subsystems such as propulsion, aerodynamics, structure, propellant management and the thermal protection system. The comparison of different potential propellants is made across the different return options in order to assess the suitability of the various possible combinations. The fuels investigated within this paper in combination with liquid oxygen are kerosene, liquid hydrogen and methane. Two different engine cycles, namely gas-generator and staged combustion, and their impact on the launcher systems are also evaluated.

Keywords: VTVL, VTHL, reusability, booster stage

Nomenclature

Isp	Vacuum Specific Impulse	s
MR	Engine Mixture Ratio	-
T	Thrust	kN
W	(Earth) Weight	N or kN
ΔV	Velocity Increment	m/s or km/s
ϵ	Expansion ratio	

RTLS	Return To Launch Site
SART	Space Launcher Systems Analysis
VTHL	Vertical Take-off Horizontal Landing
VTVL	Vertical Take-off Vertical Landing

Acronyms/Abbreviations

asc.	Ascent
dsc.	Descent
CSG	Centre Spatial Guyanais
DRL	Downrange Landing
GLOM	Gross Lift-Off Mass
GTO	Geostationary Transfer Orbit
IAC	In-Air-Capturing
LCH4	Liquid Methane
LC3H8	Liquid Propane
LH2	Liquid Hydrogen
LOx	Liquid Oxygen
MCC	Main Combustion Chamber
MR	Mixture Ratio
NBP	Normal Boiling Point
RCS	Reaction Control System

1. Introduction

While reusability in space transport can have a strong impact on the costs and thus competitiveness of space launchers, the historic Space Shuttle has also shown that this impact does not necessarily have to be positive if the refurbishment costs cannot be kept low.

Nonetheless, the recent successes of SpaceX (with Falcon 9 and Falcon Heavy) and Blue Origin (New Shepard) in landing, recovering and reusing their respective booster stages by means of retropropulsion have shown the possibility of developing, producing and operating reusable launchers at low launch service costs. Hence, the investigation of the different methods of reusability has become of essential importance for Europe in order to keep up with the evolving launch market.

In order to assess the possibility of employing reusable booster stages in future European launchers the system study ENTRAIN (European Next Reusable

Ariane) is being conducted within the X-TRAS project of the DLR. The final goal is the determination of the impact of the different return methods on a technical, operational and economical level and the identification of their potential for a future European launch system. The ENTRAIN study itself is separated into two phases: Within the first phase, shown herein, a selection of designs is investigated parametrically in order to assess the impacts of propellant, staging, engine cycle and return method. Based on the results of this phase two configurations will be selected and investigated in detail in the second phase.

This paper will show and discuss selected results obtained within the first phase of ENTRAIN. While prior papers have only compared different VTVL boosters to each other [1][2][3] and future papers will focus on the comparison of different VTHL boosters to each other [4], this paper will focus on the comparison of winged and non-winged stages and thus will limit itself to showing cases that are as comparable as possible. Nonetheless, the following section 1.1 will elaborate on the logic of the entire study.

It is also worth mentioning that in parallel to the ENTRAIN system study DLR is developing two flight demonstrators to increase its knowledge related to reusable vehicles. One of the demonstrators is dedicated to VTHL vehicle return flight and is called ReFEx for Reusability Flight Experiment [17][18]; the second demonstrator represents VTVL missions and is called CALLISTO (Cooperative Action Leading to Launcher Innovation in Stage Toss-back Operations) [15][16]. Flights and operations of both vehicles should help verify and refine assumptions used in the frame of ENTRAIN and thus improve the quality of the analysis.

1.1. Study Logic and High Level-Assumptions

As mentioned above, the ENTRAIN study is divided into two phases: the first part focuses on comparing VTVL and VTHL launchers while optimizing the launch systems to a comparative level in order to avoid distortions by different optimization levels. Therefore, the same high-level requirements and assumptions are used for both return methods. The high-level requirements and considered propellant combinations, engine cycles and stagings are the following:

- 7000 kg + 500 kg margin, payload to GTO of 250 km x 35786 km x 6° (standard Ariane 5 GTO) via a LEO parking orbit of 140 km x 330 km x 6° (see section 2.5 for details)
- Launch from CSG, Kourou
- TSTO: Two Stage to Orbit

- Same propellant combination in both stages
- Same engines in both stages with exception of different nozzle expansion ratios
- Engine Cycles: Gas Generator (GG) and Staged Combustion (SC)
- Return modes:
 - VTVL with retropropulsion landing on downrange barge (DRL) or with return-to-launch-site (RTLS)
 - VTHL with In-Air-Capturing (IAC) or autonomous return to launch site (Flyback)
- 2nd stage Δv of 6.2 km/s, 6.6 km/s, 7.0 km/s, 7.6 km/s
- Propellant Combinations: LOX/LH2, LOX/LCH4, LOX/LC3H8, LOX/RP-1 and subcooled variations

In order to keep the total number of configurations at a feasible level a “loop zero” was performed with structural index curves based on historic launchers. The method and the results for the VTVL configurations are shown in [3]. Based on these results the field was narrowed especially with regard to staging velocities. For the VTVL the RTLS option was also not further investigated since the required payload and the demanding target GTO lead to launcher sizes for which the economic relevance was clearly questionable. RTLS could however still be used for low energy missions targeting LEO or SSO.

A comparison of the common near-boiling point propellants and their subcooled and slush counterparts was also conducted for the VTVL configurations and was presented in [5]. The following deliberations, however, focus on the usually used near-boiling point propellants.

For the selected launchers a more detailed mass estimation model was established. Additional modeling of the propellant system, the engines and their cycles, the tank, structure and of several other subsystems was also performed. The resulting configurations are the focus of this paper, the methods will be described in more detail in chapter 2 and the resulting launchers will be presented in chapter 3.

At the end of this comparative phase of the ENTRAIN study, one promising concept for each the VTVL and the VTHL return method will be selected to enter the second half of the study, ENTRAIN 2, with the aim to further refine the design of the launch system and to identify required technology developments depending on the return mode. This phase will be discussed in more detail in 4.1.

2. Preliminary Design Phase Assumptions

The following section describes the models used to evaluate the reusable booster stages. If the methods used for VTHL and VTVL are different it is noted within the relevant subsystem section. Much of the following sections are based on the descriptions of the methods given in [1]. More detailed descriptions of the models used can be found in [2] and [3] for the VTVL stages and in [4] for the winged VTHL stages.

2.1. Main Propulsion Rocket Engines

In order to ensure comparability of the designed launchers, generic engines with identical baseline assumptions are needed for the systematic assessment and comparison of future RLV-stages. The selected technical characteristics of these generic engines are strongly oriented towards data of existing types as well as previous or ongoing development projects.

The two rocket engine cycles most commonly used in first or booster stages are included in the study:

- Gas-Generator-cycle
- Staged-Combustion cycle.

The main combustion chamber (MCC) pressure is commonly set to 12 MPa for the gas-generator type. This pressure is not far from the useful upper limit of this cycle but is assumed necessary to achieve sufficient performance for the RLV stages. Europe has considerable experience in this range with Vulcain 2 operating at 11.7 MPa. In the case of the staged-combustion engines, the MCC pressure is fixed at 16 MPa. This, from a Russian or US perspective, moderate value has been chosen considering the limited European experience in closed cycle high-pressure engines. Nozzle expansion ratios in the first stage are selected according to optimum performance but also requirements of safe throttled operations when landing

VTVL-stages. For the first stage of the VTVL configurations the engine is computed for expansion ratios of 20 for gas generator types and 23 for the staged-combustion variants. This value allows throttling while still retaining sufficient nozzle exit pressure to prevent flow separation within the nozzle. Since the VTHL configurations do not need throttleability, the expansion ratio is set to be 35 for gas-generator and staged combustion engines.

The upper stage engines are derived from the first stage engines with the only difference being the expansion ratio. This value was set to 120 based on the results from the first structural index design cycle showing that the comparatively lower specific impulse with respect to an expansion ratio of 180 was compensated by the mass reduction of the shorter interstage structure and nozzle. Otherwise all other engine parameters are equal to those of the first stage (chamber pressure, engine cycle, turbopump efficiencies etc.). Note that the expansion ratio value of 120 has not been optimized but constitutes a reasonable assumption, especially considering the fact that the goal of this study is to compare first stages, whereas whole launcher optimization will be performed within ENTRAIN 2.

All preliminary engine definitions have been performed by simulation of steady-state operation at 100% nominal thrust level using the DLR-tools lrp and ncc (nozzle contour calculation program) as well as the commercially available tool RPA (rocket propulsion analysis). Any potential requirements specific to transient operations or deep-throttling are not considered in this early design study. Common baseline assumption of all generic engines is a vacuum thrust in the 2200 kN-class. Although all engine mass flow rates are scaled to the required thrust level of the individual launcher configuration, the underlying assumptions on component efficiencies (e.g. turbopumps) will likely be too optimistic for much smaller engines. Turbine entry temperature (TET) is set around 750 K and kept in all cases below 800 K to be compatible with the increased

Table 1: LOX/RP-1, LOX/LC4H8, LOX/LCH4 and LOX/LH2 engine data. The first stage engines with the lower expansion ratio are used within the VTVL launchers.

Propellants	LOX/RP-1			LOX/LCH4			LOX/LH2					
Stage	1st		2nd	1st		2nd	1st				2nd	
Cycle	GG	GG	GG	GG	GG	GG	GG	GG	SC	SC	GG	SC
ϵ	20	35	120	20	35	120	20	35	23	35	120	120
Engine MR [-]	2.25	2.25	2.25	2.5	2.5	2.5	6	6	6	6	6	6
Sea level Isp [s]	279	267	-	289	276	-	366	351	394	386	-	-
Vacuum Isp [s]	310	320	338	320	331	348	406	418	428	434	440	457
Engine T/W [-]	112	113	92	98	99	83	98	96	74	72	82	70

lifetime requirement of reusable rocket engines.

Further, all engines considered in this study are designed with regeneratively cooled combustion chambers and regenerative or dump-cooling of the down-stream nozzle extensions. The results based on these assumptions are shown in Table 1. Detailed information on the respective engine modelling is given in [2] and [9].

2.2. Structure and Propellant Supply System

The masses of most structural elements including first and second stage propellant tanks, the interstage, the second stage's front skirt and the first stage's thrustframe were sized using the DLR tool *lsap* (launcher structural analysis program). The general loads have been computed based on the GTO ascent trajectory; a number of load cases were defined to size the structure (including margins for dynamic loads). The safety factor was chosen to be 1.25, a standard value for unmanned launchers.

The propellant tanks and the second stage front skirt were designed using a conventional stringer/frame approach. The number of stringers and frames is subject to an optimization process within the tool to determine the lightest configuration possible. The tanks are separated by a common bulkhead and modelled as being manufactured of the aluminum alloy AA2219. All tanks are pressurized with 3 bar which was chosen based on previous experience with launcher design and shall be subject to optimization in the second study phase ENTRAIN 2.

The interstage was modelled as an aluminum honeycomb structure with carbon fiber outer layers and was also sized within the tool *lsap*. The fairing mass was calculated by scaling the mass of the Ariane 5 fairing with the respective surface area.

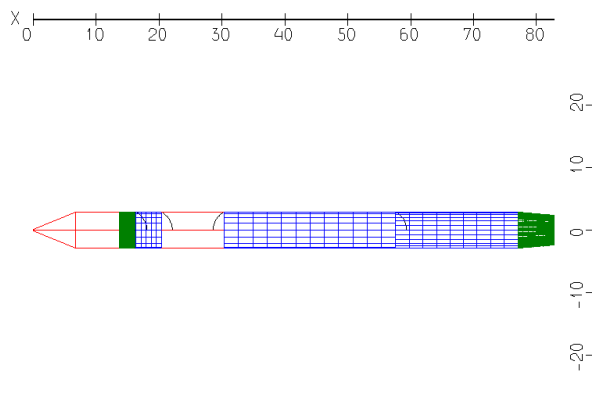


Fig. 1: Structure model of the preliminary launchers

The first stage's thrustframe was modelled as a stringed/frame-stiffened conical structure with the

rocket body diameter at the front end and 0.5 m of diameter less at its rear end (see Fig. 1 for details). The thrustframe is covered by a skirt acting as protection against the aerodynamic and aero-thermal loads during launch and re-entry.

In Fig. 1 the structural model of the preliminary hydrogen launcher is shown. The green-colored portions are the front and rear skirt, the portions colored in red represent the composite structures (interstage and fairing), the blue portions represent the tanks and the black lines display the outline of the tank domes and common bulkheads.

The propellant supply system including feedlines, fill/drainlines and the pressurization system was modelled using the DLR-tool *pmp*. This program is able to evaluate the respective masses for these systems by simulating the propellant and pressure gas flow throughout the entire mission and thus sizing the required hardware.



Fig. 2: Model of the propellant management system for a generic LOX/LH2 launcher

The model of the propellant system is shown in Fig. 2. The depicted propellant system is that of the hydrogen launcher which will be presented in detail in section 3. The LOX tanks are colored in blue, the LH2 tanks in red. The first stage main propellant feedline is branched into several smaller feedlines to individually supply the engines at the rear end of the launcher. The pressurization lines run along the backside of the tanks and supply the LOX tank with gaseous oxygen and the LH2 tank with gaseous hydrogen. The hydrocarbon launcher LOX tanks are pressurized with gaseous oxygen as well. However, while the methane tank is pressurized with gaseous methane, the RP-1 tank is pressurized with helium which is stored in separate pressure tanks.

The *pmp* tool also calculates the mass of the cryogenic insulation of the tanks. It is important to note that insulation was only considered a necessity in the case of LOX/LH2 launchers due to the low temperature of LH2. In the case of hydrocarbon launchers no insulation is used, since it adds mass and it is technically feasible to fly cryogenic propellants without insulation (e.g. Falcon 9 with LOX/RP-1).

2.3. Recovery Hardware

All investigated stages have to be equipped with additional hardware in order to enable their safe reentry and return. Since the VTVL and VTHL configurations employ different methods to achieve this goal, their recovery hardware differs completely.

2.3.1. Recovery Hardware for VTVL

The attitude of the VTVL stages has to be controlled in all phases of the return flight. During exoatmospheric flight comparatively small conventional RCS (Reaction Control System) with small thrusters can perform this task. During the atmospheric portion of the reentry aerodynamic control devices are necessary. Furthermore, the landing of the first stage requires some type of landing gear or legs. Currently, only two operational VTVL launchers are active, the first one being the Blue Origin's "New Shepard" suborbital launcher and the second one being the Falcon 9 first stage. The "New Shepard" launcher uses conventional fins, such as so-called "wedge fins", to control the aerodynamic forces during descent whereas the Falcon 9 uses grid fins.

Since the Falcon 9 first stage mass is more in the order of the proposed launchers than the New Shepard's mass, the ENTRAIN launchers' fins and landing legs mass was calculated by scaling the respective mass estimates of the Falcon 9 with the ENTRAIN launchers' dry mass. The masses of the Falcon 9 recovery hardware such as grid fin and landing leg masses were estimated using in-house tools and reverse engineering [10]. This scaling method will be replaced by a proper mass calculation using CFD calculations for the grid fins and landing dynamics and structural design considerations for the landing legs in the second part of the ENTRAIN study (see section 4.1). However, it is not decided yet if the VTVL launcher investigated in the second phase of the study will be equipped with grid fins or conventional fins.

Furthermore, the ENTRAIN launchers are equipped with a 2 cm thick layer of cork at the baseplate acting as a thermal protection system. The thickness was kept constant for all launchers since the re-entry occurs with more or less similar maximum heatflux. However, the dimensions of this TPS are difficult to determine since little to no actual data of the aerothermodynamic behavior during a first stage suborbital re-entry are available. Hence, the TPS thickness is a first guess that shall be reevaluated and thoroughly investigated in the second part of the study by means of CFD calculations such as shown in [11].

2.3.2. Recovery Hardware for VTHL

As for the VTVL stages described above, the attitude control for the exoatmospheric portion of the first stage flight is assured through RCS thrusters whereas the atmospheric flight with sufficient dynamic pressure is controlled with aerodynamic control surfaces. The VTHL first stages are equipped with a single delta wing with a leading edge sweep of 40° and a trailing edge sweep of 0° as well as a V-tail, wing flaps (ailerons) and a body flap. For the main wing a

RAE2822 airfoil is used both at wing root and wing tip. The size of the wing – root and tip chord length as well as span – is sized as a function of stage length. The wing flaps have a constant chord length along the span which in turn is a function of the overall wing root chord. The body flap serves two purposes: on the one hand it can be deflected (downward) to trim the vehicle; on the other hand it serves as a protection of the first stage engines against the thermal loads during reentry. Its span is equal to the stage diameter whereas the length is determined by the first stage engine length. In contrast to the structural segments of the fuselage the main wing and aerodynamic control surfaces structural weights cannot be calculated using the tool *Isap*. Their masses are estimated using empirical methods implemented in the DLR-tool *stsm* (Space Transport System Mass Estimation). Parameters relevant to the applied methods are mechanical loads, surface area, span length, wing thickness and stage dry mass.

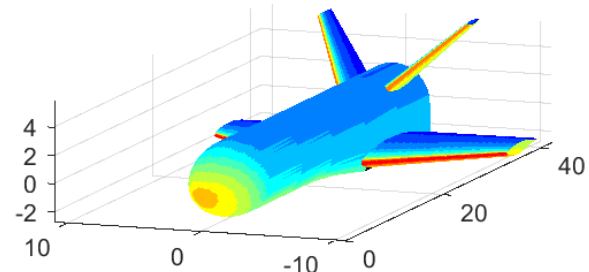


Fig. 3: Example of temperature distribution on VTHL first stage

The Thermal Protection System (TPS) is a crucial part of the recovery hardware for VTHL configurations. In the frame of the current study the mass of the thermal protection system is estimated based on the selected TPS materials and the thermal loads experienced during atmospheric reentry. Fig. 3 shows a qualitative temperature distribution and the general VTHL first stage layout. The VTHL first stages discussed in this work employ TPS materials such as space shuttle type Flexible Reusable Surface Insulation (FRSI), Tailorable Advanced Blanket Insulation (TABI) and ceramic tiles. A more detailed description of these materials can be found in [12]. The external thermal loads are determined by the DLR tool *HOTSOSE*, an aerothermodynamic code for the hypersonic regime. Following the determination of the external loads and the definition of an allowable temperature below the thermal protection the TPS thickness and mass is iteratively calculated assuming one-dimensional heat transfer.

2.4. Mass Model

The masses not already modelled by the structural analysis or other dedicated subsystem-analysis tools (for example the engines or the TPS) were modelled using the DLR-tool *stsm*. This program uses empirical estimation formulas based on historical data to calculate the masses of some structural elements such as the rear skirt and subsystem masses such as engine equipment (including engine controllers and wiring), electrics and harness. Several subsystems were scaled with Ariane 6 subsystem masses such as the power system and batteries, the pyro stage – and fairing separation system, avionics, the RCS and the payload adapter.

All first stage structural parts, subsystems and TPS were subjected to a mass margin of 14% due to the fact that the mass estimation for RLVs is of higher uncertainty than that of an ELV launcher. The mass margin on the main engines was set to 12%. All second stage elements were subjected to a mass margin of 10%.

2.5. Ascent and Descent Trajectories

While the ascent for all configurations was modelled and optimized with similar methods and thus can be presented jointly in the following sections the different re-entry methods necessitate the separate descent sections 2.5.2 and 2.5.3.

2.5.1. Ascent

The ascent for the configurations presented in [1] and [3] was optimized assuming a direct insertion into the target GTO. However, due to the comparatively short total burn time of the two-staged launchers the argument of perigee was far from 0° or 180°. Those values are necessary for the final insertion into a geostationary orbit (GEO) performed by the payload itself. In order to address this issue, the ascent for the launchers shown hereafter was performed in two separate phases. First the second stage is inserted into a parking orbit of approximately 140 x 330 km. When the second stage passes over the equator a second burn is used to bring the payload onto a GTO with its perigee and apogee directly above the equator. While this ascent strategy does necessitate a reignitable second stage engine, this requirement is automatically fulfilled, at least for the VTVL-configurations: The engine shared by both stages, as described in section 2.1, has to be reignitable in order to allow the re-entry and landing of the first stage. However, propellant has to be reserved for the additional chill-down of the second stage engine. It was assumed that the equivalent of 2 seconds of burn time would be needed for this chill-down procedure.

2.5.2. Descent – VTVL

The first stage is supposed to land on a droneship or a similar floating device in the Atlantic Ocean. The descent trajectory was optimized with respect to the minimum propellant required without violation of the boundaries. No specific landing site coordinates are defined so the optimization tool can find the optimal landing site for each launcher and separation velocity. Furthermore, specific landing conditions are prescribed that have to be fulfilled:

- Landing Flight Path Angle: $90^\circ \pm 2^\circ$
- Landing Velocity: 0 m/s – max. 2.5 m/s
- Landing Altitude: 0 m \pm 10m

The low parking orbit leads to comparatively shallow ascent trajectories and thus to low flight path angles at stage separation. This has large benefits for the propellant mass needed for the re-entry burn since the shallower re-entry trajectory allows the reduction of more velocity through aerodynamic forces without violating the trajectory constraints. These constraints were derived from structural calculations performed with the tool *lsap* [13] and from loads derived from SpaceX descent trajectories [6]:

- Dynamic pressure of < 200 kPa
- Estimated heat flux of < 200 kW/m² with respect to a nose radius of 0.5 m
- Lateral acceleration of < 3g

2.5.3. Descent – VTHL

In contrast to the VTVL and VTHL ascent trajectories no reentry trajectory optimization is done for the VTHL reusable first stages. The VTHL reentry trajectories are calculated using quasi-optimal flight control methods. An integral law is used to control normal acceleration and to determine angle of attack histories. Equations of motion in four degrees of freedom (translation and pitch rotation) are solved. Bank angle is varied to initiate a turn and achieve the desired heading towards the launch site. The beginning of the turn is chosen in order to not violate mechanical and thermal loads constraints during reentry. A more detailed description is given in [19]. The reentry flight constraints and targeted final conditions are summarized below:

- Normal acceleration n_z : < 4 g
- Final altitude: < 10 km
- Final Mach: subsonic

Although not a constraint within the applied control logic, stagnation point heat flux and dynamic pressure are minimized by variation of relevant parameters as e.g. angle of attack maximum and minimum values and the time of the turn manoeuvre initialisation. It has to be emphasized that ascent and descent trajectories for VTHL configurations are closely interconnected and an ascent trajectory optimization focusing on payload performance only might lead to very high aerothermal loads during the reentry of the first stage. In particular the flight path angle at first stage separation has a significant influence on stagnation point heat flux during reentry. Thus it was attempted to minimize flight path angle at separation by constraining the pitch rate in the pitch kick phase during ascent, an essential optimization parameter for ascent trajectories. This approach is limited by the fact that with decreasing separation flight path angles an increase in dynamic pressure occurs which might be problematic from the stage separation point of view. Within this study a dynamic pressure of less than 1 kPa is considered to be tolerable at separation.

In terms of reusable first stage return and landing mode two options would exist for the VTHL stages presented; a downrange landing or “In-air-capturing” (IAC) of the stage and tow-back to launch site by a carrier aircraft. No analysis of the capturing process and the tow-back flight is performed within this study. Nonetheless since “In-air-capturing” would be the preferred option due to the lack of “natural” downrange landing sites a short summary of this method is given below.

A schematic of the reusable stage's full operational circle is shown in Fig. 4. At the launcher's lift-off the capturing aircraft is waiting at a downrange rendezvous area. After its MECO the reusable winged stage is separated from the rest of the launch vehicle and afterwards follows a ballistic trajectory until reaching denser atmospheric layers. Below 20 km altitude it decelerates to subsonic velocity and rapidly loses altitude in a gliding flight path. At this point a reusable returning stage usually has to initiate the final landing approach or has to ignite its secondary propulsion system.

Within the in-air-capturing method, the reusable stage is awaited by an adequately equipped large capturing aircraft (most likely fully automatic and unmanned), offering sufficient thrust capability to tow a winged launcher stage with restrained lift to drag ratio. Both vehicles have the same heading still on different flight levels. The reusable unpowered stage is approaching the airliner from above with a higher initial velocity and a steeper flight path, actively controlled by aerodynamic braking. The time window to successfully

perform the capturing process is dependent on the performed flight strategy of both vehicles. The entire manoeuvre is fully subsonic in an altitude range from around 8000 m to 2000 m [20]. The upper constraint is set by the requirement to reach full aerodynamic controllability of the winged stage. After successfully connecting both vehicles, the winged reusable stage is towed by the large carrier aircraft back to the launch site. Close to the airfield, the stage is released, and autonomously glides to earth without further propulsion.

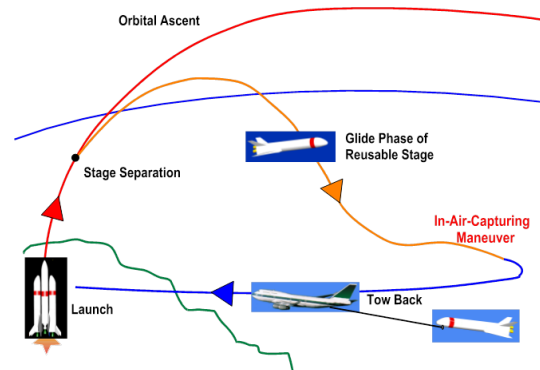


Fig. 4: Schematic of the proposed in-air capturing method

The simulation of the selected flight strategy and the applied control algorithms show a robust behaviour of the reusable stage in reaching reach the capturing aircraft. In the nominal case the approach manoeuvre of both vehicles requires active control only by the gliding stage. Simulations with reasonable assumptions for mass and aerodynamic quality show that a minimum distance below 200 m between RLV and aircraft can be maintained for up to two minutes [20]. The most promising capturing technique is using an aerodynamically controlled capturing device (ACCD), showing the best performance and lowest risk [20].

DLR is currently preparing for flight testing the “in-air-capturing”-method on a laboratory scale by using two fully autonomous vehicles [14].

3. Results & Discussion

Based on the abovementioned assumptions and methods eight configurations will be presented and discussed hereafter:

- For each VTVL (DRL) and VTHL (IAC)
 - LOX/LH2 with GG-engine
 - LOX/LH2 with SC-engine
 - LOX/LCH4 with GG-engine
 - LOX/RP1 with GG-engine

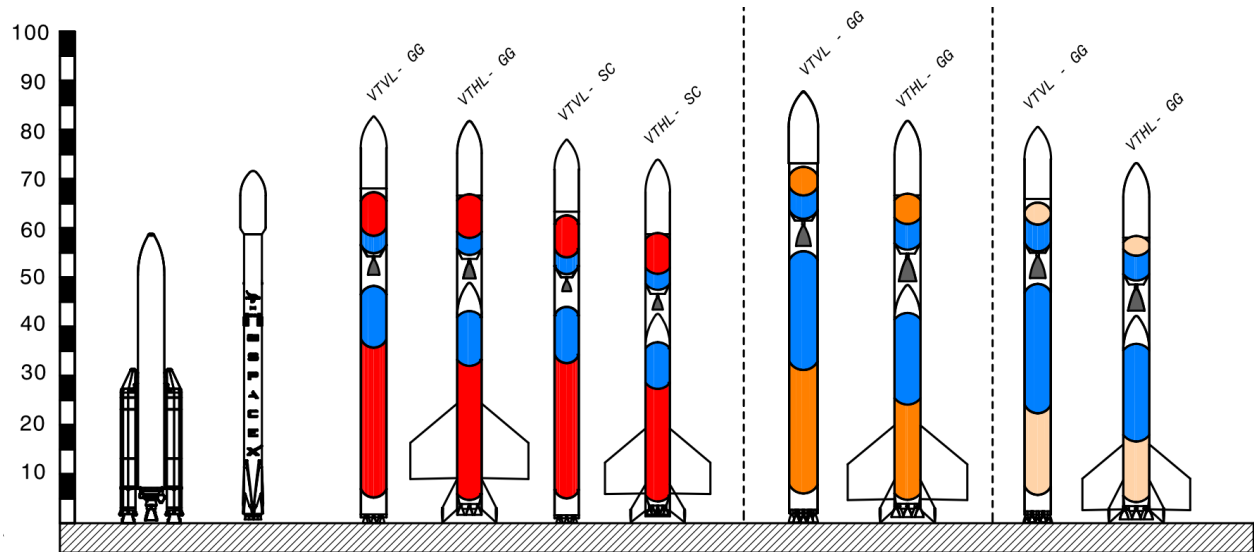


Fig. 3: Size and geometry of the investigated Semi-RLV configurations. Blue tanks contain LOX, red tanks LH2, orange tanks LCH4 and peach colored tanks RP1. GG stands for gas generator cycle engines while SC stands for staged combustion cycle engines

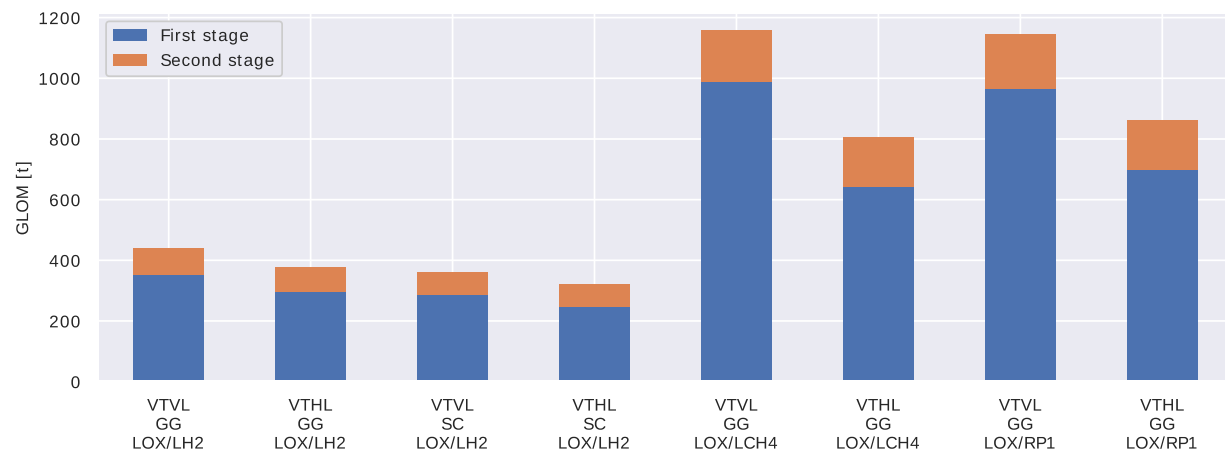


Fig. 4: GLOW of all investigated Semi-RLV configurations

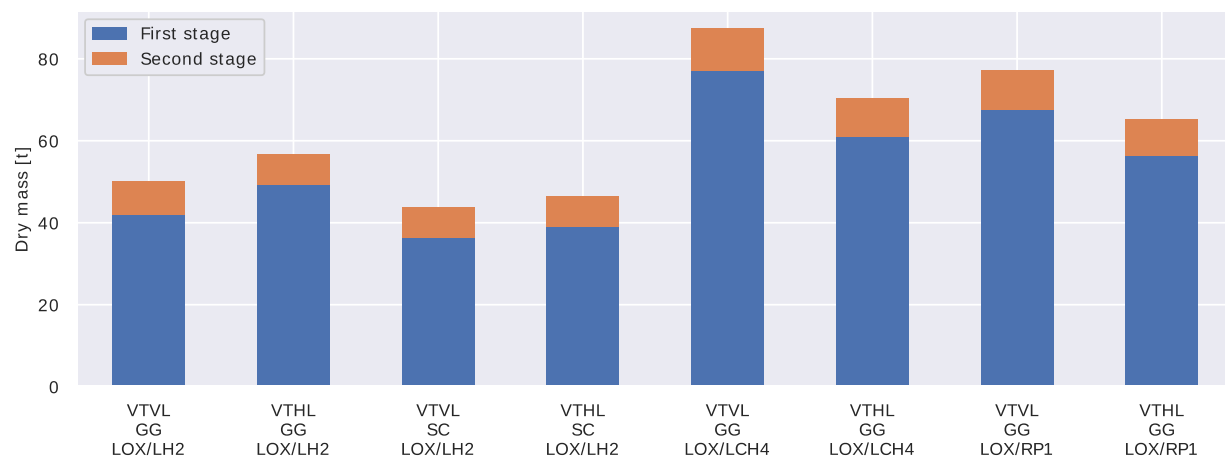


Fig. 5: Dry mass of all investigated Semi-RLV configurations

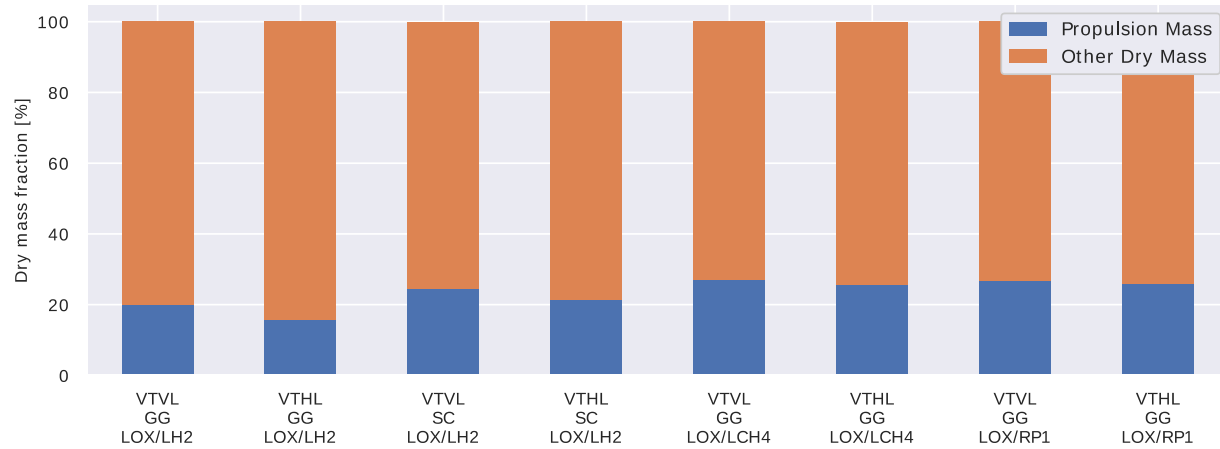


Fig. 6: Breakdown of first stage dry mass into propulsion and other subsystems of all the investigated Semi-RLV configurations

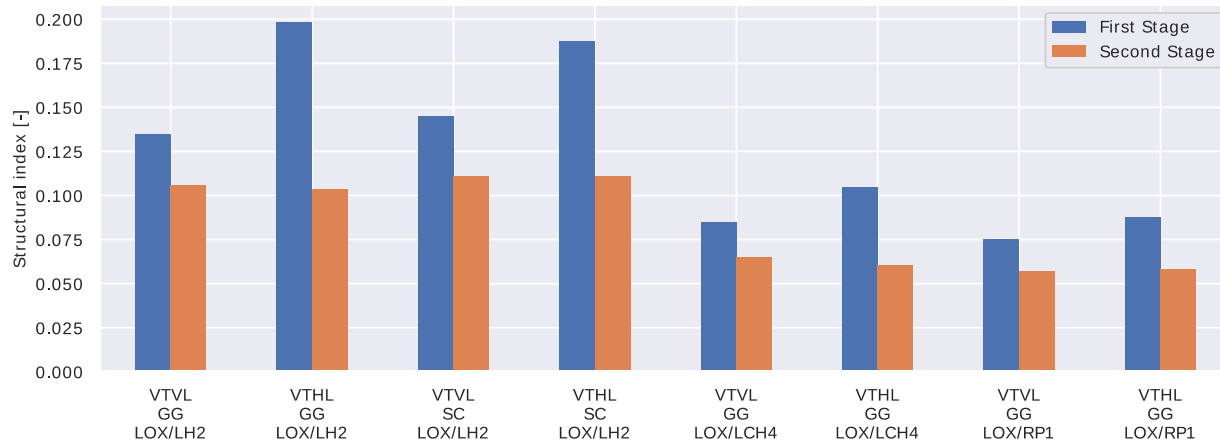


Fig. 7: Structural indices of all the investigated Semi-RLV configurations

For all of these configurations the upper stage delivers approximately 7 km/s of Δv . Additional stagings were considered but are not shown here in order to focus on the comparison of the return modes. However, they are shown in [1] and [4]. This chapter contains the results as well as the discussion of these. Section 3.1 contains the results for all launchers but special comparisons with regard to propellant and engine choice will be highlighted in the sections 3.2 and 3.3.

3.1. General Remarks

3.1.1. Metrics of Comparison

While the main metric of comparison for the various options for the next generation of European launchers should to be the final cost of placing a specific payload into a designated orbit, the estimation of the cost of

launcher or technology development programs is a notoriously difficult and inexact undertaking. This is especially true for reusable launch vehicles since no experience with these types of systems exists in Europe and even in the USA experience is limited. Nonetheless, some preliminary cost estimations are underway and will be published at a later date.

Instead, the following sections will focus on the GLOM and total dry mass of the investigated configurations. Of these two the dry mass can be seen as a better indicator of final cost since the GLOM is dominated by the propellant mass. For liquid propellant stages the propellant costs, however, are usually a negligible fraction of the total cost.

3.1.2. Comparability of Return Mode

While the two return modes of the launchers shown above are sufficiently similar some context is necessary

before diving into the following results and their discussion.

Both return methods rely on a second vehicle towing the reusable stage back to the launch site. For the winged IAC method this necessitates the use of a sizable airplane while for the ballistic DRL method a barge and, depending on the exact architecture, a towboat are necessary. In both cases the first stage has to successfully reenter the atmosphere and land autonomously. In the case of the IAC method the towing phase takes place before the landing. With DRL the towing takes place after the landing. Thus, from a performance perspective, both methods are deemed comparable. The main differences lie in the operational scenario and the economic impact.

It should be noted that even though SpaceX has successfully implemented the DRL method, neither method has been used in Europe up to this point in time. Demonstrator projects will be needed to improve the understanding and subsequently modelling of both methods in order to arrive at dependable designs. Within the Framework of the Horizon 2020 the DLR will be conducting the FALCon [14] project in which the in-air capturing maneuver will be tested with UAVs. While the upcoming reusable VTVL demonstrator Callisto [15], [16] will be significantly smaller than an operational booster stager, it will cover and demonstrate a large number of the maneuvers necessary and provide valuable insight into the operation of such a vehicle.

3.1.3. Size and Geometry

In Fig. 3 sketches of the geometry of the Semi-RLV's are shown alongside the Ariane 5 and the Falcon 9. At a first glance it obvious that some boundary conditions of the parametric study, namely the high Δv target orbit, the two stage architecture and the reusability of the first stage, lead to very large configurations. The largest launchers are fueled with LCH₄ and the smallest are the hydrogen fueled launchers with staged combustion engines. The hydrogen launchers with gas generator engines are similar in size to the RP1 configurations.

3.1.4. Mass

The gross lift-off masses of all configurations are shown in Fig. 4. At first glance the large difference between the hydrocarbon launchers and their hydrogen counterparts is obvious. The scale of this difference is magnified by the very demanding reference mission into GTO including reuse of the first stage. For less demanding target orbits the comparison would yield different results.

Generally, the gross mass of the VTHL configurations is lower than the equivalent VTVL

configurations. The magnitude of this difference is quite small for the hydrogen launchers. However, for the hydrocarbons the variation is substantial. The difference appears especially large considering that the upper stages have very similar total mass and that thus the entire difference results from the first stage.

Fig. 5 displays the dry mass of the aforementioned configurations. In this case the comparison of VTVL and VTHL is not as straightforward since VTVL configurations actually have both the highest and lowest dry mass of all cases. The main influence seems to be the propellant choice and will thus be discussed in section 3.3.1.

A breakdown of the dry mass of the first stage into the propulsion system and the other subsystems is shown in Fig. 6. The mass of the propulsion system is singled out because it has special significance for the comparison, as the propulsion system usually poses a large fraction of dry mass and a larger fraction of the costs. In general, it can be seen that the propulsion subsystem of the VTVL stages are a higher fraction of their stage dry mass. For the VTHL the fraction of the propulsion system is kept lower by the additional dry mass added for recovery. This difference is larger for the hydrogen stages than for the hydrocarbon stages.

3.1.5. Structural Index

The structural indices of all abovementioned configurations are shown in Fig. 7. It is defined here as

$$SI = \frac{m_{dry}}{m_{prop}}$$

It includes the engine dry mass as well as, for the upper stage, the fairing. Contrary to "usual" ELV-type launchers the structural index of the upper stage is in all cases lower than the first stage. This difference is caused by the additional recovery hardware needed in the first stage. As expected the winged first stages have the highest structural indexes.

In theory the upper stages for the individual propellant and engine choice should be identical for VTHL or VTVL versions, because the upper stage is not directly impacted by the return mode of the first stage. However, some secondary factors influence the second stage. For example the diameter of the launcher is strongly influenced by the first stage and in turn affects the structural mass of the second stage. Another example is the exact thrust to weight ratio of the second stage: Since the upper stage engine has the same mass flow as the first stage engines and the thrust-to-weight ratio is kept at 1.4 at lift-off for all configurations, the main parameter for influencing the thrust of the upper stage is the number of engines on the first stage. Since this number obviously has to be an integer, the size and weight of the upper stage engine cannot be kept exactly identical for all configurations. Nonetheless, even with

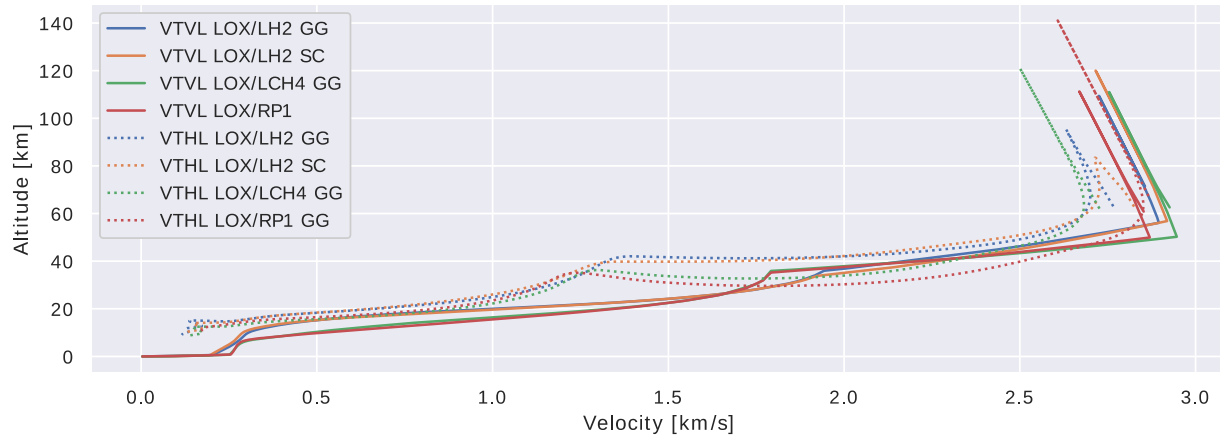


Fig. 8: Comparison of descent trajectories: Altitude over velocity

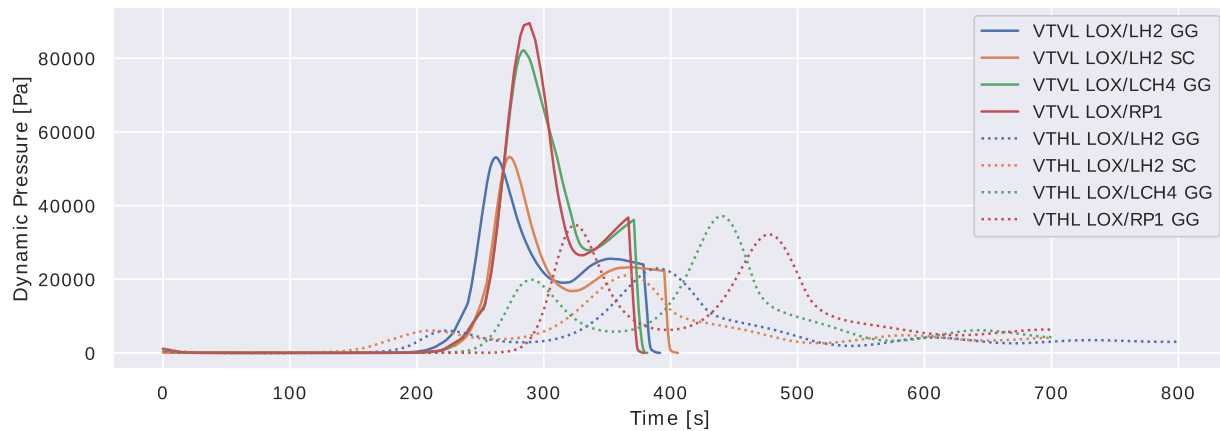


Fig. 9: Comparison of descent trajectories: Dynamic pressure over time

these effects the upper stages of the equivalent VTVL and VTHL launchers have sufficiently similar structural indexes to not unduly influence the comparison of the first stages.

3.1.6. Descent Trajectory

The re-entry profile for all discussed launchers is shown in Fig. 8. Since all launchers discussed here have an upper stage that delivers a Δv of 7 km/s the separation conditions are similar. The difference in initial conditions visible in Fig. 8 is caused by the accuracy used during iterating the launcher designs: The demanded Δv of the upper stage of 7 km/s was required to be met within ± 100 m/s in order to keep the number of design iterations to a feasible level. The fact that the VTHL configurations seem to generally separate at lower velocities than the VTVL originates from the different design procedures used. This results in some upper stages delivering slightly more than 7 km/s while others deliver slightly less while all remaining within the allowed margin. Thus, all initial reentry conditions

are within ± 100 m/s of the reference value and can be compared to each other. The difference in Fig. 8 appears larger but this is caused by variations in flight path angle and altitude at separation, which both have an impact on the Δv that can effectively be delivered by the upper stage.

Until the end of the reentry burn of the VTVL configurations the reentry profiles of VTVL and VTHL are fairly similar. After that point the different aerodynamic properties lead to different reentry trajectories. In both cases the fuel and the thus resulting ballistic coefficient has a major impact on the descent trajectory. This influence will be discussed in 3.3.3.

Fig. 9 displays the dynamic pressure experienced by all launchers during the duration of their descent trajectories. As can be clearly seen, the descent of the winged stages takes place over a significantly longer period of time. This is caused by the wings being used to prolong the reentry in order to keep the maximal loads small. For the VTVL the impact of propellant choice is clearly visible in the maximum dynamic pressure experienced. This will be further expanded

upon in section 3.3.3. Based on the analysis made in [13] it is not estimated that any level of dynamic pressure will lead to a substantial structural mass increase for the VTVL configurations, since the stage is comparatively light during these phases and the loads acting upon the launcher structure are not larger than during ascent flight.

3.2. Influence of Engine Cycle

The following subsections and figures focus on the impact of using gas-generator or staged combustion cycle engines for the cycle. Since both options were only considered for the hydrogen launchers, the comparison is necessarily limited to that fuel.

3.2.1. Mass

GLOM and dry mass for the hydrogen launchers with GG and SC cycle engine are shown in Fig. 10 and Fig. 11 respectively. Naturally, the higher specific impulse delivered by the staged-combustion cycle engines enables smaller launchers to deliver the required payload to orbit. The GLOM reduction is higher for the VTVL than for the VTHL launchers. It makes sense, as the VTVL is relying both for ascent and descent on its engines. Having highly performant engines is consequently an important factor for VTVLs.

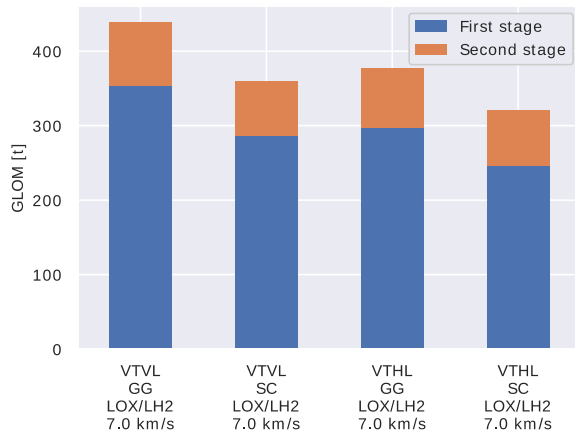


Fig. 10: Comparison of GLOM for different engine cycles

For the dry mass the trend is the same: For both VTVL and VTHL the dry mass is substantially lower for the launchers with staged-combustion engines.

When comparing the percentage of dry mass taken up by the propulsion system in Fig. 6 it is visible that for the configurations with staged-combustion engines these pose a larger fraction of the overall dry mass. This is consistent with the lower thrust-to-weight ratio of this

engine type, as shown in Table 1 and the fact that with higher specific impulse less propellant and thus stage mass is needed for the same performance. Consequently, one could conclude that using stage combustion engines is advantageous for reusable launchers since it leads to lower masses and possibly, using dry mass as a rough cost indicator, to lower costs. However, the development and qualification process of a staged combustion engine is usually more complex, time-consuming and costly compared to gas generator engines.

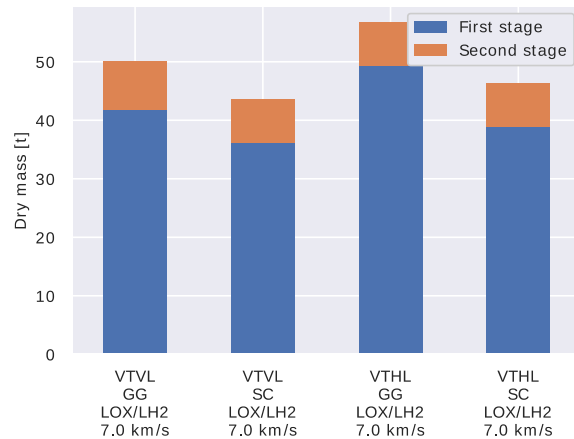


Fig. 11: Comparison of dry mass for different engine cycles

3.2.2. Structural Index

As can be seen in Fig. 12 in all cases except one, the structural index for the stages using staged-combustion engines is higher than for their equivalent with gas-generator cycle engines. This is caused by the decrease in stage size enabled by the higher specific impulse in addition to the slightly higher mass of the staged combustion cycle engines, since their thrust-to-weight ratio is lower. Since some subsystems do not scale linearly with the propellant mass this leads to higher structural indexes. The one exception is the first stage of the VTHL configuration. This can be explained by the fact that the respective stage has a different diameter to length ratio than the other stages in order to keep the diameter above 5 m so that the fairing is large enough to accommodate typical GTO payloads. This thicker design has advantages from a structural perspective and leads to a lower structural index even though less propellant is loaded. While this inadvertent optimization does distort the comparison it is not easily avoidable in order to allow the demanded diameter of the fairing. While it is possible, and frequently done (e.g. Falcon 9), to choose a fairing diameter larger than the stage diameter, this would also negatively impact the comparability of the stages.

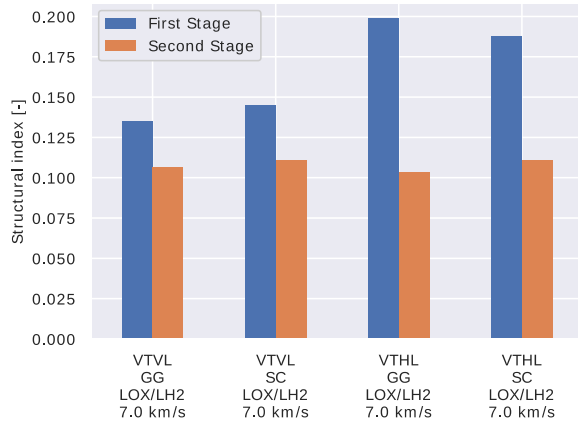


Fig. 12: Comparison of structural index for different engine cycles

3.2.3. Descent Trajectory

As can be seen in Fig. 8 and Fig. 9 the descent trajectories of the staged-combustion and gas-generator stages are very similar for both VTVL and VTHL. The choice of the propellant type has a far larger impact and will be discussed within the following sections.

3.3. Influence of Fuel Type

Within this paper three fuel options are shown and discussed: hydrogen, methane and RP1. Earlier publications also include some other propellant options such as propane [1], [3] and explored the possibility of using subcooled or slush propellants for the VTVL configurations [5]. It is noteworthy that propane and subcooled propellants display interesting advantages. However, this paper is limited to hydrogen, methane and RP1 since launchers for both VTVL and VTHL were designed for these propellants thus allowing this comparison. This does not imply that they are the better options.

Generally, the choice of fuel impacts two major parameters: specific impulse and bulk propellant density. Other aspects of the launcher are, of course, also impacted by the fuel choice but not to the same degree as the aforementioned two. The consequences for the specific impulse are already listed in the engine model description in section 2.1. The bulk propellant density dictates the necessary tank volume and thus influences the structural index. In both these categories methane and RP1 are fairly similar, especially when compared to the extreme case hydrogen (highest specific impulse and lowest density). Thus in the following deliberations they will often be grouped together as hydrocarbons.

3.3.1. Mass

Fig. 13 shows the GLOM of all launchers with gas-generator engines in order to allow a focus on the impact of the propellant choice on both VTVL and VTHL configurations. While for hydrogen as a fuel the GLOM of the VTVL and VTHL configurations are similar, for the hydrocarbons the VTHL stages are significantly smaller.

This trend remains true even when considering the dry mass, shown in Fig. 14. While here the VTVL generally fare better than when comparing GLOM, thanks to the lower structural index of the first stage, for the hydrocarbons their dry mass still is higher than the equivalent VTHL dry mass. Only in the case of the hydrogen-fueled configurations the dry mass of the VTVL versions is actually lower than their VTHL equivalents. As mentioned before, this is mostly caused by the lower specific impulse of the hydrocarbon fuels, which is especially detrimental for the VTVL cases since they use propellant for accelerating (ascent) and decelerating (reentry) and thus have to deliver a higher total Δv over both phases of the mission.

When considering the dry mass fraction of the propulsion subsystem as shown in Fig. 6 it becomes apparent that the hydrocarbon stages need comparatively larger engines even though the engines themselves have higher thrust-to-weight ratios. This is caused by the much higher GLOM of these stages: In order to achieve the desired thrust-to-weight ratio of the entire system at launch a larger propulsion subsystem relative to the dry mass is needed.

It should be noted that the use of methane and kerosene as fuel leads to similar GLOMs for both VTHL and VTVL. When comparing the dry mass however, the kerosene fueled launchers are noticeably lighter than their methane-fueled counterparts. Apparently the ca. 10 s higher specific impulse is not sufficient to compensate the higher structural index of the methane stages.

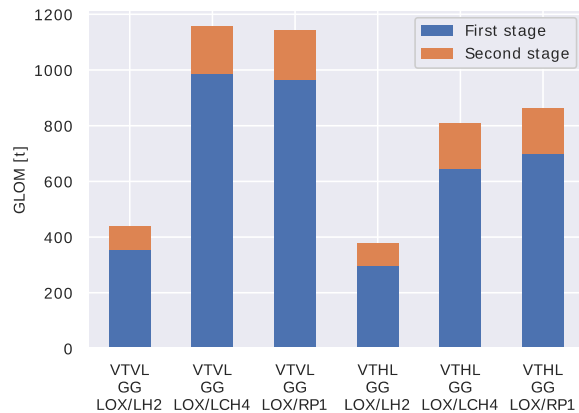


Fig. 13: Comparison of GLOM for different fuel choices

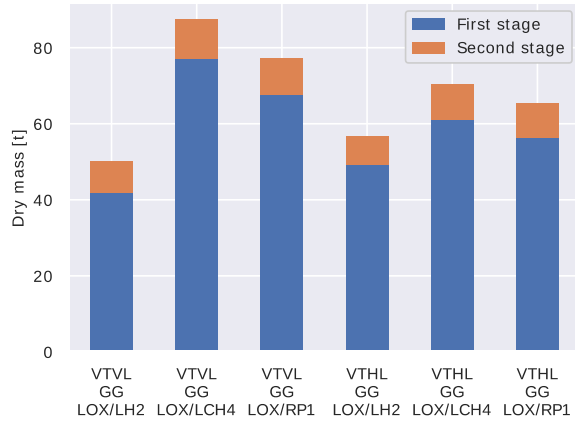


Fig. 14: Comparison of dry mass for different fuel choices

3.3.2. Structural Index

As expected the structural indexes for the hydrocarbon stages are significantly lower than for the hydrogen stages with the use of RP1 leading to the lowest structural indexes. While for both VTVL and VTHL hydrogen leads to the highest structural indexes, the difference to the hydrocarbons is especially large for the VTHL case. This is caused by the scaling of the wings with the length of the first stage, which leads to large wings for the hydrogen-fueled stages, relative to the dry mass.

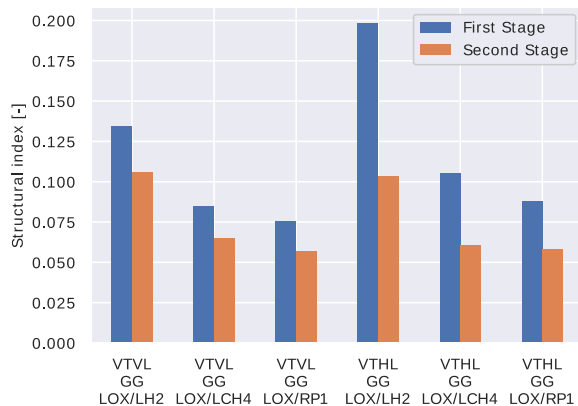


Fig. 15: Comparison of structural index for different fuel choices

3.3.3. Descent Trajectory

As can be seen in both Fig. 8 and Fig. 9 the choice of fuel has a major impact on the descent trajectory for both VTHL and VTVL stages. The underlying reason is that higher propellant densities result in stages with higher ballistic coefficients, even if the stage does not actually contain any propellant anymore. In general, a

lower ballistic coefficient is advantageous for both reentry types since more velocity can be shed through aerodynamic deceleration, but the type of impact is different.

For the VTVL configurations a higher ballistic coefficient necessitates a more extensive reentry burn. In Fig. 8 it can be seen that the reentry burn of the hydrocarbon stages reduces the speed of the stage down to approximately 1.7 km/s while the reentry burn of the hydrogen stages can be terminated at about 1.9 km/s without violating the trajectory constraints. The missing Δv is achieved through additional aero-braking, which is more efficient because of the lower ballistic coefficient of the hydrogen stages.

The maximum dynamic pressure experienced along the trajectory is substantially higher for the hydrocarbon stages. Due to the aforementioned higher ballistic coefficients this does not effectively result in more aerodynamic deceleration.

For the VTHL stages the higher ballistic coefficients of the hydrocarbons stages necessitates entering deeper in the atmosphere before generating sufficient lift to slow the fall of stage. Another effect amplifies this: The VTHL stages with lower specific impulse have significantly shorter burn times and thus have to perform a much steeper ascent in order to reach a permissible dynamic pressure (1 kPa) at separation. The shorter burn time results from the fact that thanks to the lower specific impulse a larger fraction of the total propellant loading has to be burned per time unit in order to achieve the required thrust-to-weight ratio during ascent. These steeper trajectories lead to higher flight path angles at stage separation and for reentry into the atmosphere. Both effects lead to higher heat loads during reentry for the hydrocarbons stages, which in turn lead to a comparatively heavy TPS. As another side effect the dynamic pressure experienced is also higher than for the hydrogen versions.

The VTVL stages are not affected by the second effect because their engines have lower expansion ratios, as noted in section 2.1, and thus the mass flow rates needed for the required thrust at lift-off are lower. This causes their burn times to be sufficiently long that the ascent optimization is not impacted by the dynamic pressure at separation constraint and they all separate at very similar flight path angles.

4. Conclusions

This paper compared VTVL and VTHL configurations for different propellant combinations and engine cycles. The comparison showed that the general size of hydrogen as well as hydrocarbon launchers with reusable booster stages increases significantly compared to common ELV's. However, the GLOMs of the hydrogen fuelled launchers are in the same order of

magnitude as operational ELV-vehicles such as Ariane 5.

As expected, using LOX/LH2 results in the lightest launchers followed by the hydrocarbon fuelled launchers. With regard to the GLOM the difference between the two shown hydrocarbon options, LCH4 and RP1, is comparatively small. The difference becomes significantly larger when comparing the dry mass, where the use of RP1 always leads to lower total dry mass of the launcher. The difference between the hydrogen and hydrocarbon configurations was especially large for the VTVL configurations. The designs with methane as fuel always have the highest dry mass of all investigated propellants. The effect of propellant choice has a significant impact on both re-entry and recovery methods that is not limited to the two major factors specific impulse and propellant density but can be noticed in secondary effects that appear after evaluating the entire mission and the affected subsystems.

In general, the VTHL configurations have lower GLOM than the equivalent VTVL configurations. The difference is especially large when using a hydrocarbon fuel.

When considering the dry mass, the configurations with a VTVL first stage and hydrogen as fuel appear to be the most attractive solution. However the distance to the equivalent VTHL stages is not large enough to discount them, especially considering the many uncertainties in this type of parametric investigations. With hydrocarbon fuels however, the VTHL configurations have significantly lower dry mass than the equivalent VTVL configurations.

The use of staged combustion engines leads to large GLOM and dry mass reduction for either re-entry option but more so for the VTVL configurations.

4.1. Outlook

On the basis of the results shown herein two configurations will be selected for further study in the ENTRAIN 2 study. Therein the comparability of the configurations is no longer paramount but instead each system is optimized to fulfill a range of mission scenarios. The main goal of this phase is the evaluation of detailed realistic designs and the identification and investigation of technological challenges on subsystem level.

The focus on two configurations will allow the use of high-fidelity and computationally intensive tools developed by DLR. This will enable the investigation of some critical phenomena that cannot be assessed with the preliminary methods, e.g. the controllability or the

integration of the recovery hardware (TPS, wings, landing gear etc.) into the core stage structure.

5. Acknowledgements

The authors would like to acknowledge the contribution of everyone in the X-TRAS team involved in design process of the herein presented launchers.

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